ELECTROSTATIC DISCHARGE ANALYSES FOR SPACECRAFT IN GEOSYNCHRONOUS ORBIT

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ABSTRACT

Surface charging of spacecraft materials in the geosynchronous orbit environment is caused primarily by electrons with energies oscillating between 1 and 50 KeV during magnetospheric substorms. The potentials reached during charging events depends on the total turret balance among secondary-electron emission, backscattered electron flux, and photoelectron currents. Differential charging occurs when parts of a spacecraft are charged at different negative potentials relative to each other. In this type of charging strong local electric fields develop. When the electric fields exceed critical values Electrostatic Discharges (ESD) can occur causing EM I noise and posing potential threats to spacecraft instruments, A review is first conducted of the physics involved in the charging and discharging mechanisms in spacecraft at geosynchronous orbit. An electromagnetic computational technique., based on the method of moments, is then explored to analyze Electromagnetic Interference (EMI) effects as a result of ESD events.

1.0 INTRODUCTION

This paper deals with the interactions effects caused by the. space charged particle environment: satellite charging and discharge. The physics of spacecraft charging processes is described first and it will be followed by modeling approaches used to analyze Electrostatic Discharge (ESD) events in spacecraft systems, specially the use of the Method of Moments for determining the induced current distribution resulting from ESD events.

There are two types of charging processes in a spacecraft. The first type is surface charging, Surface charging, is the result of accumulation of electrostatic charge on satellite exterior surfaces due to interactions between the surfaces and the plasma environment. Surface charging can be further divided into two classes: 1) absolute charging and 2) differential charging. In absolute charging the entire spacecraft potential relative to the ambient plasma is charged uniformly, Aside from effects on plasma instruments, this class of charging is typically of little concern to the design engineers. In differential charging parts of the satellite charge to different potentials relative to each other. In this type of charging, strong local electric fields may exist. Differential charging can occur due to the following: a) the plasma flux to the satellite may be spatially nonuniform so that electrical isolated surfaces are charged differently; b) local material properties may vary around a satellite; c) the surface potentials between the sunlit and shaded parts are different due to photoelectron emission, d) odd configurations (e.g cavities) may cause strange charging behaviors. Differential surface. charging could lead to electrostatic discharge or arcing between the satellite surface when differential charging generates an electric field that exceeds the dielectric breakdown strength of the material. Hence, it is of serious concern to the design engineers.

Surface charging/discharge is most likely to occur in an environment in which an intense flux of energetic electrons (in the kiloelectron volt range) is presented, such as in geosynchronous and polar orbit and during periods of magnetospheric storms,

The second type of charging is internal charging. Internal charging is caused by electrons and ions of high enough energy to pass through the spacecraft surface and deposit charge on and within materials inside the satellite. Internal discharge occur when floating metal or dielectrics collect enough charge so that the electric field exceeds the breakdown strength from the point of the deposited charge to a nearby point. Internal discharges are important when the spacecraft is expected to operate in an environment where there is large fluences ($> 10^{11}/cm^2$) of energetic electrons (energy range 300 keV-5 MeV). In this paper we will not consider internal charging in more details.

When a discharge occurs (ESD), charge is rapidly released from the discharge site until the potential gradient no longer exists. The most severe effect of electrostatic discharge is that the electromagnetic interference (EMI) generated during a discharge can cause temporary or permanent interruption in the operation of cm-board electronic systems. Arcing is also associated with dielectric breakdown, hence could damage the dielect ric thermal con troll coating of the satellite surface, Opt ical surfaces can be contaminated from redeposition of materials ejected due 10 arcs. Hence, it is important to investi gate: a) possible discharge paths that can affect sensitive electronics and b) possible magnitude of the ESD induced currents in the previously identified paths.

2.0 SURFACE CHARGING

Most generally, the current collected by a spacecraft and the. amount of the charge buildup are related through

$$\frac{\partial Q}{\partial t} + \int \vec{J}\cdot d\vec{s} = 0 \qquad (1)$$

where Q is the charge on the spacecraft, J is the. incident current density from the plasma, and the integration is over the spacecraft surface. The current collected by a satellite is a function of its geometry, surface voltage and surface propert ies. When a total current balance is achieved, we have

$$\frac{\partial Q}{\partial t} = 0$$

and the system is in equilibrium state.

2.1 Charging Current Sources

In order to calculate the charging potentials, one needs to analyze the charging current sources. 'I'here are seven major charging mechanisms which can deposit charge on a surface. These mechanisms are shown in Figure 1

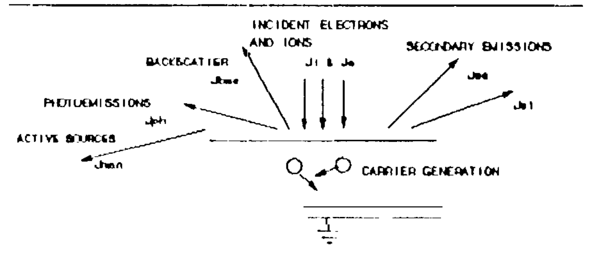

FIGURE 1 ELECTRON AND IONS INTERACTIONS RELATED TO CHARGE BUILDUP IN INSULATORS

a) Ambient plasma fluxes J_i and J_e:

The major natural source of potentials of 10 kV or higher is the ambient plasma. Garret and DeForest [3] have shown that the geosynchronous plasma environment can be modeled by a two Maxwell-Boltzmann distribution functions.

$$f_k(v) = n_{k1}\left(\frac{m_k}{2\pi KT_{k1}}\right)^{3/2} \exp\left(\frac{m_k v^2}{2KT_{k1}}\right)^{3/2} \cdot$$

$$n_{k2}\left(\frac{m_k}{2\,pi\,KT_{k2}}\right)^{3/2} \exp\left(\frac{-m_k v^2}{2KT_{k2}}\right) \qquad (2)$$

where the subscript k stands for the species (k= e for electron and k== i for ion), K is the Boltzmann constant, and m_k, T_k, and n_k are the mass, characteristic temperature, and number density of the species k respectively. This representation in most cases tits the measurements quite adequately over the energy range of importance to spacecraft charging. In Table 1 we list the fitted plasma parameters for average environments measured by ATS-5 and SCATHA [1]. The plasma fluxes collected by a spacecraft surface may be estimated by applying the so called "probe theory". For a spherical body and a Maxwell-Boltzmann distribution, the first order current fluxes are given by

$$J_e = \sum_{j=1}^{2} J^0_{ej} \exp\left(\frac{q\phi}{KT_{ej}}\right) \qquad \phi < 0 \qquad (3)$$

$$J^* = \sum_{j-1}^{2} J^0_{ej} \exp\left(1 \; \frac{q\phi}{KT_{ej}}\right) \qquad \phi > 0 \qquad (4)$$

$$J_i = \sum_{j-1}^{2} J^0_{ij} \exp\left(1 - \frac{q\phi}{KT_{ij}}\right) \qquad \phi < 0 \qquad (5)$$

$$J_i = \sum_{j=1}^{2} J^0_{ij} \exp\left(-\frac{q\phi}{KT_{ij}}\right) \qquad \phi > 0 \qquad (6)$$

where. $J_{ej}^0 = (qn_{ej}/2)\sqrt{(2KT_{ej}/\pi m_e)}$, $J_{ij}^0 = (qn_{ij}/2)\sqrt{(2KT_{ij}/\pi m_i)}$, and Φ is the voltage of the body. The values for J_{ej}^0 and J_{ij}^0 are also in Table 1. For design purposes, one should use the "worst-case" environment rather than the average environment to calculate the spacecraft potential. In Table 2 we list the worst case environments measured by SCATHA, which may be used as a design case environment. More examples of test environments can be found in Appendix A of Purvis et al [1].

b) Photoelectron currents J_{ph}:

The photoelectron current from a surface is a complex function of surface material, potential, solar flux, and solar incident angle. Photoelectron currents are important in space because many materials have large photoelectron yields at extremes wavelengths (< 2000A) where the solar spectrum also has significant energy.

c) Secondary electron flux J_{se} and J_{si}:

Secondary electrons are electrons emitted as a result of **energy deposition by incident electrons and** ions. It is a complex function of incident particle energy, the emissions spectrum of secondary electrons, secondary electron yield, and the distribution function of incident particles. As a gross approximation, for a negatively charged aluminum surface the secondary flux due to electrons J_{se} is 40 % of the incident flux while that due to protons J_{si} is more like 80 % to 300 % [2].

$n(cm^{-3})$	$T(eV)$	$J(A/m^2)$
	ATS-5	
n_{e1} = 0.578	T_{e1} = 277	J^0_{e1} = 2.58 x 10^{-7}
n_{e2} = 0,215	T_{e2} = 7040	J^0_{e2} = 4.83 x 10'
n_{i1} = 0.75	T_{i1} = 300	J^0_{i1} = 8.11 x 10” Q
n_{i2} = 0.61	T_{i2} = 14000	J^0_{i2} = 4.51 x 10^{-8}
	sCATSA	
n_{e1} = 0.10	T_{e1} = 5s0	J^0_{e1} = 4.9 x 10”
n_{e2} = 0.31	T_{e2} = 8680	J^0_{e2} = 7.7 x 10^{-7}
n_{i1} = 0.19	T_{i1} = 800	J^0_{i1} = 3.36 x 10^{-9}
n_{i2} = 0.39	T_{i2} = 15800	J^0_{i2} = 3.06 x 10^{-8}

Table 1. Average plasma environment from ATS-5 and SCATSA

$n(cm^{-3})$	$T(eV)$	$J(A/m^2)$
n_{e1} = 1.8	T_{e1} = 600	J^0_{e1} = 1.18 x 10^{-6}
n_{e2} = 3.3	T_{e2} = 25500	J^0_{e2} = 1.4 x 10^{-5}
n_{i1} = 2.0	T_{i1} = 350	J^0_{i1} = 2.34 x 10^{-8}
n_{i2} = 3.3	T_{i2} = 25100	J^0_{i2} = 3.26 x 10^{-7}

Table 2. Worst case environment from SCATHA

d) Backscattered electron flux J_{bse}:

Backscattered electrons are those ambient electrons reflected back from the spacecraft surface (usually with some energy loss). For negative surface potentials, J_{bse} is roughly 20 % of the incident flux [3].

e) Active current sources J_{man}:

Sometimes a spacecraft also curries artificial current sources. For instances, electron and ion beams have been used to probe the environment and to control the surface potential. Typically these beams are ejected with energy in the kilovolt range and with current magnitude between milliamperes and amperes.

2.2 Equilibrium Surface Potential

The calculation of the surface potential involves first determining the current to the satellite surface, and then, finding a potential so that the total current balance is achieved. For a uniformly conducting spacecraft, the equilibrium surface potential is given by the solutions of the total current balance equation:

$$\sum I_{source}(\phi) = I_e(\phi) + I_i(\phi) + I_{se}(\phi) + I_{si}(\phi) + I_{bse}(\phi) + I_{ph}(\phi) + I_{man}(\phi) = 0 \quad (7)$$

In the above equation all ion currents have positive values while all electron current have a negative value.

For a shadowed passive surface, the dominant charging mechanism is often the ambient plasma flux, J_e and J_i. Since the electrons are much more mobile than the ions over the same energy range, the current balance for a shaded surface is often achieved at a very negative potential. As an example, let us calculate the equilibrium potential for a small satellite (r< 10m) with uniform aluminum surface in eclipse. For this case, J_e and J_i may be calculated using equations (3) through (6). As a simple approximation, J_{si}, J_{se}, and J_{bse} may be estimated with typical values $J_{si}/J_i \approx 3$, $J_{se}/J_e \approx 0.4$, and $J_{bse}/J_e \approx 0.8$. We also have $J_{ph} = 0$ for a shadowed surface. Solving Φ_w from equation (7), we find the potential to be $\Phi_w = -T_e$ where the electron temperature T_e is in electron volts. Hence, for the first order in eclipse, the spacecraft potential is approximately equal to the plasma temperature. Typical range of the equilibrium potential for a shadowed surface is from -300 V to -20000 V.

For a sunlit surface, the photoelectron J_{ph} is normally large enough to balance the incident electron current J_e. As a result, the equilibrium potential for a sunlit surface is typically a few volts positive. Similarly, for a highly emissive surface, the incoming electrons and ions may cause the secondary electrons and backscattered electrons to balance the incident electron flux. Hence, the equilibrium potential for highly emissive surfaces is also at a few volts around the zero potential. Differential charging can thus easily develop between a shadowed dielectric surface (which is at a high negative potential) and a sunlit surface or highly emissive surface (which is at near-zero positive potential).

Usually the current-voltage characteristic for a surface is monotonic and crosses the zero current level $\Sigma I(\Phi) = 0$ only once. The equilibrium potential for such a surface is stable because after a slight perturbation, the system moves to restore its equilibrium state. However, it has been shown that under certain magnetospheric conditions, a surface's current-voltage characteristics may have an abnormal behavior.

Under these conditions, the surface-plasma system may adjust itself in more than one way to satisfy the current balance condition. Mathematically this means multiple solutions of@ could exist for the equation $\Sigma I(\Phi) = 0$.

3.0 SURFACE DISCHARGE

3.1 Discharge Model

Given that differential surface charging can occur in the geosynchronous environment, it can also lead to electrical discharges or arcs on the satellite surface.

The general picture of dielectric. discharge is that incident electrons penetrate the material and form a space charge layer at a depth on the order of microns below the surface. As the electric field within the material increases, a critical value is eventually reached and the breakdown of material occurs accompanied by material vaporization and ionization. Discharges can occur from the dielectric to space, from metal to metal, and between metal and dielectric. The discharge process is complicated because of its dependence on material properties, especially surface composition and microscopic imperfect ions such as cracks, holes, etc. and on the macroscopic configuration such as grounding arrangements and proximity of other materials. While no totally consistent theory exist regarding the discharge onset mechanism at present, recent theoretical studies have proposed a few microscopic models for arcing onset mechanism. For example, Park et al [4] attributes the arcing onset to the Malter effect at a thin dielectric layer on the conductor surface by the following scenario: 1) a dielectric impurity layer is formed on the conductor surface, 2) ions attracted by the negative biased conductor are accumulated on the dielectric layer and enhance. the electric field inside the layer, 3) electrons are emitted from the metal-dielectric interface via Fowler-Nordheim field emission, 4) these emitted electrons produce positive charges within the dielectric layer throgh ionization collisions, 5) as the electrons are emitted from the dielectric layer into space, the positve charges are left behind and thus further enhence the electric field inside the layer, and 6) in this model the breakdown condition is given by the expression $\exp(\alpha_d d_d) P_{d\text{-}v} > 1$, where α_d is the ionization rate per unit distance inside the layer, d_d is the layer thickness, and $P_{d\text{-}v}$ is the probability that electrons are emitted from the dielectric vacuun interface. When the above condition is met, the above scenario will result in a field and emission current runaway, thus leads to an arcing onset. It has been commonly agreed by the engineering community that if either of the following two criteria is exceeded, discharge can occur [2]: 1) dielectric surface voltages are greater than 500 V positive relative to an adjacent exposed conductor, or 2) the interface between a dielectric and an exposed conductor has an electric field larger than 10^5 V/m.

3.2 Discharge EMI and Modeling

When an electrostatic discharge occurs, the release of the stored charge. creates a discharge current. Characterizing the discharge EMI has been done mainly by experimental means because of the lack of an analytical model. The discharge generated EMI has two components: a) conducted emissions and b) radiated emissions. When a discharge occurs, charge is blown off from the dielectric surface which induces a replacement current to flow in the satellite structure. The conducted emissions are characterized by this replacement current. Since the replacement current is proportional to the rate of the surface voltage change, a rapid surface potential change induces noise in circuits through capacitive coupling. The transient current that can be coupled into a nearby circuit can be estimated from

$$i \propto C\frac{d\phi_w}{dt} \propto I_{replace} \qquad (9)$$

The discharge current can also induce an inductively coupled signal into the victim circuit, which can be estimated from

$$v \propto L\frac{dI_{replace}}{dt} \qquad (10)$$

where C and L are the capacitance and inductance between the victim circuit and the discharge surface respectively. Typical peak arcing current generated by a dielectric material of 50 cm^2 is in the range of 50-100 Amps. The typical rise time of a discharge current pulse is about 10 ns or less. Hence, the corresponding $dI_{replace}/dt$ is 10^{11} A/see. Therefore, the inductively coupled signal can be of significant amplitude. The discharge current pulse also produces RF noise, which is characterized by a radiated RF spectrum. Attempts have been made to model this radiated RF spectrum as a elementary dipole over a ground plane [5].

One aspect of the radiated RF spectrum that has not been addressed is the radiated emissions caused by transient current as it propagates through several paths in the spacecraft structure. In essence, several elements of the spacecraft structure become "temporary antennas" as they become conductors of this transient current. This "antenna behavior" can be analyzed using integral or differential forms of Maxwell's equations. The objective is to estimate the radiated fields as a function of the current distribution along the conductive paths followed by the discharge current. The difficulty of this type of problem requires the use of advance computational electromagnetic techniques. Even then, the topology of this problem can be quite complex. A useful electromagnetic computational technique that can be used in these analyses is the Method of Moments (MOM) [6-8]. The MOM is based on an integral form of Maxwell's equations and allows the modeling of a current distribution on a discretize conductive path. From the known current distribution the radiated electromagnetic fields can then be found.

As a simple example consider a discharge event between a solar array and a shielded signal cable as shown in Figure 2. The discharge process in Figure 2 can be modeled, asa first approximate ion, by the lumped parameter network representation of Figure 3. Each element needs to he calculated or estimated [9]. The initial charged voltage potential generated across the glass covered solar array (i. e voltage potential across C_6 in Figure 3) can be obtained from a known space plasma environment using equation 7 (a modeling too] that performs such calculations numerically is given in [10]). More simplistic analytical models can also be used [9].

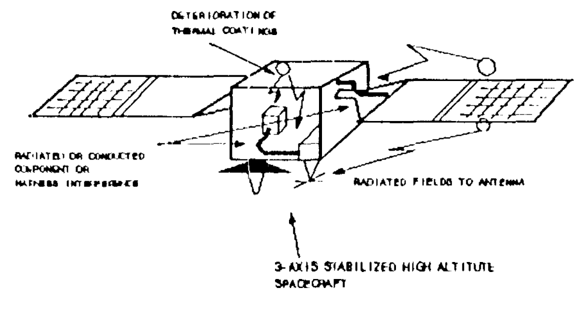

FIGURE 2. ESD EVENTS DUE TO SPACECRAFT CHARGING

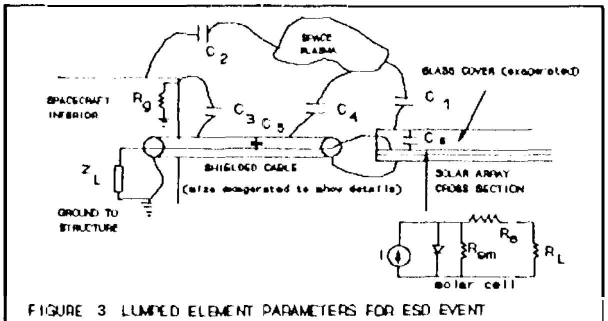

FIGURE 3 LUMPED ELEMENT PARAMETERS FOR ESD EVENT BETWEEN SOLAR ARRAY AND CABLE

A circuit representation of the lumped parameter model is shown in Figure 4. Worst case transient analyses can be simulated using SPICE-like codes in order to calculate the conducted current (i. e conducted noise emissions) al points of interests (e. g noise current at load Z_1). Of importance is the need to calculate the ESD-induced noise current distribution on the shield as a result of the ESD event, This noise current adds to the return current of the signal cable. As shown in Figure 5, the shield is grounded using a pigtail connection (shield-to-ground) for the return current. The radiated noise emissions generated by the pigtail, due to the added noise current, can be analyzed using the MOM. Figure 5 shows the thin-wire modeling of the shield, inner conductor, and pigtail. Since MOM is a frequency domain technique a frequency dependent driving-source voltage must be modeled as input to the radiating model (shield/inner conductor/pigtail). This source modeling can be done by calculating the transient voltage across a coupling, impedance term and then calculating the Fourier Transform to determine magnitude spect rum (see reference 9 for example),

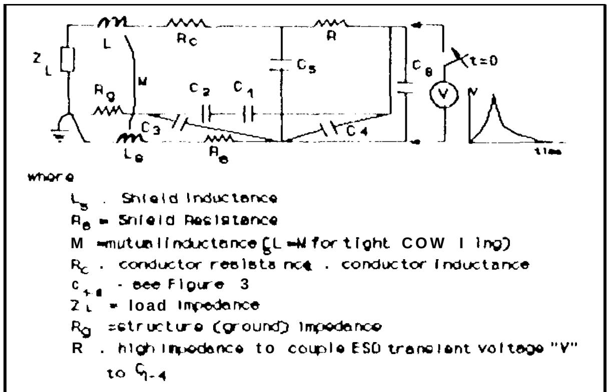

FIGURE 4. CIRCUIT REPRESENTATION OF SOLAR ARRAY TO SPACECRAFT ESD EVENT

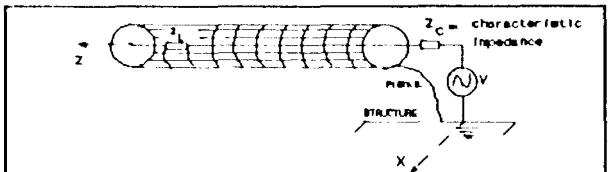

FIGURE 5. METHOD OF MOMENT MODELING OF SHIELDED CABLE

4.0 CONCLUSION

'l'his paper has outlined some of the basic physics involved in the electrostatic charging of spacecraft surface-s. This work was also extended to describe the physics of electrostatic discharge and how the induced charge deposition can generate conducted noise emissions and radiated emissions. For radiated emissions resulting from transient current propagating on spacecraft structures, a novel technique was discussed to calculate the electromagnetic fields from a modeled current (distribution using the Method of Moments.

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